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FIXED-BASE-SIMULATOR STUDY OF ABILITY OF PILOTS TO PERFORM SOFT LUNAR LANDINGS BY USING A SIMPLIFIED GUIDANCE TECHNIQUE

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SUMMARY

A six-degree-of-freedom fixed-base-simulator study has been conducted of the ability of pilots to perform soft lunar landings by using a simplified guidance technique to deorbit from a synchronous transfer orbit and to place the landing vehicle in a position from which a vertical descent to touchdown in a specified area can be accomplished. The pilot had control of vehicle thrust along the longitudinal axis and of attitude through an acceleration command system. No automatic damping or control was assumed. The general piloting procedure consisted of maintaining a constant thrust angle with respect to the orbiting command module until nearly zero velocity was attained at an altitude of approximately 5000 ft (1524 m). A vertical descent was then made to the lunar surface. Initially, a nominal trajectory was flown for which the deorbiting procedure was specified; subsequently, several off-nominal trajectories were flown.

The results of the investigation indicated that the pilot's use of the simplified guidance technique with rather crude thrust-angle measurements resulted in placing the vehicle in a position from which soft landings in the desired lunar area could consistently be made. The characteristic velocity required for piloted landings was within about 10 percent of that required for a perfectly flown nominal trajectory.

INTRODUCTION

Simplified guidance techniques for manually controlling various phases of a lunar mission are currently of interest. These manual procedures can serve as backup guidance modes or, if sufficiently precise, might be considered as primary control modes. Such procedures should require a minimum of equipment.

In the analytical study of reference 1 the orbiting command module was used as a reference for thrust-vector orientation in performing the landing phase of a lunar mission. The study indicated that maintaining a constant angle between the thrust vector and the line of sight to the orbiting command module resulted in placing the landing vehicle about 5000 ft (1524 m) above the desired landing site with nearly zero velocity. From this point a vertical descent to the lunar surface can be made (ref. 2).

The present investigation was performed to determine the ability of pilots to use the simplified guidance technique of reference 1 to deorbit from a synchronous transfer orbit and to place the landing vehicle in such a position that a subsequent descent to the lunar surface can be accomplished at a given downrange position. In actual practice the terminal phase probably would be a flare maneuver during which visual cues would be obtained by sighting on the lunar terrain. Since proper visual cues for the flare maneuver could not be generated with the available simulator, the terminal phase consisted of a vertical descent to the lunar surface by use of position and velocity information from instrument readings. The present fixed-base-simulator study permitted six degrees of freedom of the vehicle, and the equations of motion were solved by using a combination of a digital differential analyzer and an analog computer operating in real time. The pilot closed the control loop and had direct input into the force and moment equations. The simulated flights were initiated at the pericynthion (altitude, 50 000 ft or 15.24 km) of a synchronous transfer orbit from the 486 000-foot (approximately 80 n. mi. or 148.1328 km) altitude circular orbit of the command module.

SYMBOLS

Measurements for this investigation were taken in the U.S. Customary Units but are also given parenthetically in the International System of Units (SI). (See ref. 3.)

F rocket thrust along body z-axis, positive in -z-direction, lb (N)

g_e acceleration at surface of earth due to gravitational attraction, 32.2 ft/sec^2 (9.814 m/sec^2)

g_m acceleration at lunar surface due to gravitational attraction, 5.32 ft/sec^2 (1.6215 m/sec^2)

h altitude above lunar surface, ft (m)

I_{sp} specific impulse, 303 sec

I_X,I_Y,I_Z moments of inertia about body X-, Y-, and Z-axes, respectively, $I_X = I_Y$, slug-ft^2 (kg-m^2)

M_X,M_Y,M_Z control moments exerted about body X-, Y-, and Z-axes, respectively, ft-lb (m-N)

m vehicle mass, slugs (kg)

$\dot{m}$ time rate of fuel consumption, slugs/sec (kg/sec)

p,q,r angular velocities about body X-, Y-, and Z-axes, respectively, radians/sec

R radial distance from center of moon, ft (m)

ΔR_a range-to-go to nominal landing site, ft (m) (see fig. 6)

R_m lunar radius, 5.702×10^6 ft (1.7379×10^6 m)

$\dot{R}$ vehicle radial velocity component, ft/sec (m/sec)

$R\dot{\theta}$ vehicle circumferential velocity component, ft/sec (m/sec)

t time, sec

ΔV characteristic velocity, $g_e I_{sp} \log_e \frac{m_o}{m}$, ft/sec (m/sec)

W_o earth weight of vehicle in orbit, $m_o g_e$, lb (N)

X,Y,Z vehicle body axes with origin located at vehicle instantaneous center of gravity and with Z-axis alined with vehicle axis of symmetry

x_i,y_i,z_i inertial reference axes with origin located at center of moon (see fig. 1)

x,y,z distances along x_i-, y_i-, and z_i-axes, respectively, ft (m)

x',y',z' reference axes parallel to inertial axes with origin located at vehicle center of gravity

Δz vehicle lateral displacement with respect to initial orbit plane, ft (m)

α angular orientation of vehicle in pitch, defined as approximate angle between local horizontal and vehicle Z-axis, radians or deg (see fig. 6)

β angle between thrust vector and line of sight to orbiting command module, positive below command module, deg (see fig. 6)

δ_F rocket throttle control displacement, radians or deg

$\delta_X,\delta_Y,\delta_Z$ control displacements which produce control moments about X-, Y-, and Z-axes, respectively, radians or deg

ψ,θ,ϕ Euler angles of rotation, radians or deg (see fig. 1)

$\bar{\psi}$ angular orientation of vehicle referred to as yaw and defined as angle between vehicle XZ-plane and trajectory plane, radians or deg

$\bar{\theta}$ angular travel over lunar surface, radians

$\bar{\phi}$ bank angle, radians or deg

Subscript:

o initial conditions

A dot over a symbol indicates differentiation with respect to time.

EQUATIONS OF MOTION

The equations of motion used in this study permitted all six rigid-body degrees of freedom of the vehicle. The three force equations were written with respect to an inertial-axis system and the three moment equations were written with respect to the body axes. (See appendix.) The inertial-axis system was a fixed-axis system with its origin at the center of the moon (fig. 1), which was assumed to be a nonrotating homogeneous sphere. The pilot closed the control loop and had direct input into the force and moment equations. Vehicle mass and moments of inertia were varied as thrust was applied to account for mass reduction during thrusting. Mass changes due to moment control were neglected because they were small in comparison with the mass change due to thrust application.

VEHICLE DESCRIPTION

The landing vehicle assumed for this study was a relatively squat body of revolution. The vehicle had a single fixed engine which provided thrust along the axis of symmetry with a maximum capability of accelerating the vehicle initial weight at $0.485g_e$ $\left(F/W_o = 0.485\right)$. Thrust was assumed throttleable from full thrust to zero thrust and was assumed to be restartable.

Moment control about the three vehicle axes was assumed to be produced by reaction jets operating in pairs to produce pure couples. The variations of the vehicle moments of inertia with vehicle mass assumed for the vehicle under consideration are shown in figure 2.

Cockpit and Controls

A photograph of the cockpit used in the study is presented in figure 3, which shows the relative positions of the pilot's chair, throttle, controls, and displays. Vehicle thrust was commanded by using the throttle located to the left of the pilot's chair. Thrust varied linearly with control displacement. Attitude control was provided through an acceleration command system by using the three-axis hand controller located to the pilot's right. Control inputs commanded by the pilot for attitude control resulted in control torques which were proportional to control deflection except for a small dead band around zero deflection (fig. 4).

Information Display

The cockpit shown in figure 3 is a general-purpose apparatus, which was used in other studies and therefore includes instrumentation not used in the present investigation. A sketch indicating the instrument display used in this study is shown in figure 5. Above the instrument panel is an oscilloscope display in which three stars and the command module are represented by dots, and the lunar horizon is depicted by a straight line. This view corresponds to that observed through a periscope looking along the axis of symmetry in the direction of thrust (fig. 6). The assumed periscope had a field of view of $\pm 30^{\circ}$ and a reticle with markings 6° apart. The use of this display made possible the estimation of vehicle attitude in pitch and yaw to within about 1°.

The oscilloscope display provided the necessary information for thrust orientation during the deorbit phase of the landing. However, a three-axis gyro-horizon was included for use during the vertical descent phase of the landing when the horizon is not visible on the oscilloscope. In addition, an α-vernier was supplied in order to permit the pilot to monitor pitch attitude very closely during the vertical descent phase. This indicator was scaled to read from 85° to 95° with a resolution of 0.2°.

Digital readout meters presenting radial velocity $\dot{R}$, circumferential velocity $R\dot{\theta}$, and altitude h were included. The resolution of the meters was 1 ft/sec (0.3048 m/sec) for radial velocity, 10 ft/sec (3.048 m/sec) for circumferential velocity, and 10 ft (3.048 m) for altitude. Circumferential velocity must be less than 10 ft/sec (3.048 m/sec) at touchdown; thus an $R\dot{\theta}$-vernier was included which had a resolution of 1 ft/sec (0.3048 m/sec). In addition, the altimeter operated inaccurately below 1000 ft (304.8 m); so that a second meter, which possessed a resolution of 10 ft (3.048 m), was required to operate below 1000 ft (304.8 m). Additional meters were included which presented out-of-plane position, out-of-plane velocity, and range-to-go to the nominal touchdown point with resolutions of 2500 ft (762 m), 5 ft/sec (1.524 m/sec), and 400 ft (121.92 m), respectively. Because of the unavailability of a completely visual simulation during the vertical descent to the lunar surface, the metered position and velocity information was used in the present investigation and these readings represent the actual values.

LUNAR LANDING TRAJECTORIES

The nominal deorbit trajectory was the constant-thrust and constant-thrust-angle approximation of the gravity-turn descent used in reference 1. The landing vehicle is assumed to be in a synchronous transfer orbit from the 80-nautical-mile-altitude circular orbit of the command module. When the landing vehicle reaches the 50 000 ft (15 240 m) pericynthion of the synchronous orbit, thrust is initiated at a level which results in a value for F/W_o of 0.485. Thrust is maintained at this level and is directed 23° (as determined through an iteration process in ref. 1) below the line of sight to the command module until the vehicle reaches approximately zero velocity at an altitude of about 5000 ft (1524 m). The pilot then varies thrust and attitude as might be required to descend vertically to a soft landing.

Off-Nominal Trajectories

The off-nominal trajectories were the result of varying the initial conditions about the nominal values. The assumed variations were ±50 ft/sec (±15.24 m/sec) in the initial velocity components and ±10 000 ft (±3048 m) in altitude. The following table lists the trajectories (including those which diverge most from the nominal trajectory) flown during the investigation:

Radial velocity, $\dot{R}$, ft/sec (m/sec)	Circumferential velocity, $R\dot{\theta}$, ft/sec (m/sec)	Altitude, h, ft (m)	Comments
0 (0)	5673 (1729.13)	50 000 (15 240)	Nominal trajectory
-50 (-15.24)	5673 (1729.13)	50 000 (15 240)	Low in $\dot{R}$
0 (0)	5623 (1713.89)	50 000 (15 240)	Low in $R\dot{\theta}$
0 (0)	5673 (1729.13)	40 000 (12 192)	Low in h
0 (0)	5673 (1729.13)	60 000 (18 288)	High in h
-50 (-15.24)	5623 (1713.89)	40 000 (12 192)	Low in $\dot{R}$, $R\dot{\theta}$, and h
50 (15.24)	5723 (1744.37)	60 000 (18 288)	High in $\dot{R}$, $R\dot{\theta}$, and h

All possible trajectories are not included as a matter of expediency.

PILOTING PROCEDURE

The manual guidance technique employed in the present investigation is designed to place the landing vehicle approximately 5000 ft (1524 m) above the desired landing site with nearly zero velocity. From this point the pilot should be able to perform a vertical descent to the lunar surface by using primarily visual information. In the absence of a completely visual simulation, metered information of vehicle position and velocity was included to permit the completion of the landing maneuver.

The general procedure was to observe vehicle velocity and altitude, which might be obtained from onboard radar or from the command module, prior to thrust initiation to determine whether the landing vehicle was on the nominal trajectory. If nominal conditions prevail, the pilot initiates thrust when the altitude reaches 50 000 ft (15 240 m) and directs thrust 23° below the line of sight to the orbiting command module. The pilot maintains this thrust level and thrust angle ($\beta = 23^{\circ}$) until zero velocity is attained at an altitude of about 5000 ft (1524 m), at which time the pilot adjusts vehicle attitude and thrust level to perform a vertical descent to the lunar surface. A sketch of the trajectory relative to the lunar surface is presented in figure 6.

If off-nominal conditions exist, the pilot must modify the landing procedure in order to perform soft landings at the desired landing site. Off-nominal values of initial circumferential velocity primarily affect landing range while initial altitude and radial-velocity errors primarily affect terminal altitude (ref. 1). With the initial acceleration due to thrust ($0.485g_e$) known, the off-nominal values of circumferential velocity can be accounted for by early or late thrust initiation. The effects of off-nominal values of altitude and radial velocity can be accounted for by applying thrust at a new thrust angle given by the following equation

$$\beta = 21.05^{\circ} + \frac{0.43h_o + 225.0\dot{R}_o}{1.1 \times 10^4} \tag{1}$$

where β is expressed in degrees, h_o in feet, and $\dot{R}_o$ in feet per second. (To express this equation in SI units requires that the constant 0.43 be changed to 1.41 with h_o in meters and the constant 225.0 be changed to 738.2 with $\dot{R}_o$ in meters per second.) This equation was obtained through the use of the results given in figures 12, 15, and 16 of reference 1 and a total differential equation:

$$\beta = \beta_o + \frac{\partial\beta}{\partial h_T}\frac{\partial h_T}{\partial h_o}\Delta h_o + \frac{\partial\beta}{\partial h_T}\frac{\partial h_T}{\partial \dot{R}_o}\Delta\dot{R}_o$$

where $\Delta h_o = h_o - h_{o,nominal}$, $\Delta\dot{R}_o = \dot{R}_o - \dot{R}_{o,nominal}$, and h_T = altitude at end of deorbit phase. In this case the pilot maintains the indicated thrust angle until the circumferential velocity is reduced to nearly zero at an altitude of approximately 5000 ft (1524 m). This point could be approximately determined by observing the lunar terrain or by using a body-fixed integrating accelerometer to determine when the desired velocity change has been applied. The pilot, using only visual information, should then be able to descend vertically to the desired landing site.

This technique for flying the off-nominal trajectories requires only the solution of the β equation and a knowledge of the initial circumferential velocity to provide deorbit capability. The necessary information can be obtained by using onboard radar or, if the onboard system fails, by using the command-module radar system. In the latter case, the command-module radar could be used to determine the velocity and altitude of the landing vehicle shortly after it has transferred to the elliptical transfer orbit. The command-module computer could then be used to predict the velocity components and altitude to solve for β and time of thrust initiation.

RESULTS AND DISCUSSION

The results of this investigation are divided into two sections; the first dealing with flights of the nominal trajectory and the second, with flights of the off-nominal trajectories.

Nominal Trajectory

The problem was initiated by applying thrust approximately at the 50 000-foot-altitude (15 240-meter-altitude) pericynthion of the synchronous transfer orbit with the vehicle thrust axis directed approximately 23° below the line of sight to the orbiting command module. Results of a typical piloted nominal trajectory are presented in figure 7. The solid line in figure 7(a) is the trace of the reference trajectory (ref. 1) and the dashed line is the trace of the piloted flight. It can be seen that the piloted trace follows the nominal path very closely throughout most of the flight. The time history of the flight (fig. 7(b)) shows that the pilot maintained a thrust angle of 23° until about 283 seconds after thrust initiation, at which time the velocity components approached zero at an altitude of about 5000 ft (1524 m). The pilot then reduced thrust and pitched the vehicle to descend vertically to terminate in a soft landing.

A summary of touchdown conditions of nominal flights is presented in figure 8. These results show that the pilots generally touched down with velocity components of less than 10 ft/sec (3.048 m/sec) and within a range of about 6000 ft (1828.8 m) of the nominal landing site. It should be noted that no attempt was made to decrease the range error during the vertical descent phase of the landing. A perfectly flown nominal trajectory requires a characteristic velocity of approximately 6120 ft/sec (1865.376 m/sec). The results of the piloted flights show characteristic velocities (an indication of fuel consumption) that are generally within about 300 ft/sec (91.44 m/sec) (about 5 percent) of that required for a perfectly flown nominal trajectory. A study of the flight records indicated that this variation in characteristic velocity is closely associated with the altitude at which the constant thrust-angle deorbit phase brings the velocity to zero. If the altitude at which the constant-thrust-angle phase is terminated is higher than the nominal value, the required characteristic velocity is, in general, greater. This increase occurs because the higher altitudes at initiation of the vertical descent phase require thrust to be applied against the gravity vector for longer periods of time. This condition is comparable to hovering and is expensive in terms of fuel consumption. The reverse situation occurs when the constant-thrust-angle phase is terminated at an altitude lower than the nominal.

Off-Nominal Trajectories

A total of six off-nominal combinations of initial conditions were considered in the investigation. Most of the flights were performed with the pilots using information given to them through the solution of the equation for the thrust angle β. The flight history of a typical off-nominal trajectory is presented in figure 9. The initial conditions for this flight were a radial velocity of -50 ft/sec (15.24 m/sec), a circumferential velocity of 5623 ft/sec (1713.89 m/sec), and an altitude of 40 000 ft (12 192 m). Thrust was initiated approximately 3 seconds later than would have been necessary for a nominal trajectory in order to account for the initial low value of circumferential velocity. In addition, the pilot pitched the vehicle so that the thrust angle β was approximately 21.5° as computed from equation (1) for the off-nominal initial conditions. The time history of the flight (fig. 9(b)) shows that the pilot maintained the computed thrust angle until about 280 seconds after thrust initiation, at which time the circumferential velocity became zero at an altitude of about 6000 ft (1828.8 m). The pilot then reduced thrust and pitched the vehicle in order to descend vertically to terminate the flight in a soft landing.

An alternate technique, in which approximate corrections for the off-nominal initial conditions were applied, was considered. In this technique it was assumed that radial velocity could be monitored as it would be with an onboard radar system. Off-nominal values of circumferential velocity were accounted for in a manner similar to the previous technique by initiating thrust earlier or later than required for the nominal trajectory. Thrust, however, was initially directed at an angle which immediately brought off-nominal values of initial radial velocity to the nominal value. The pilot then pitched the vehicle to a thrust angle β that exceeded the nominal value by about 0.5° for every 10 000 ft (3048 M) that the initial altitude exceeded the nominal value. Conversely, the thrust angle β was less than the nominal thrust angle by 0.5° for every 10 000 ft (3048 m) that the initial altitude was less than the nominal value. The pilot maintained this thrust angle until circumferential velocity was reduced to nearly zero at an altitude of approximately 5000 ft (1524 m) and then descended vertically to the lunar surface. There was no discernible difference between the touchdown conditions attained by the two methods. (See fig. 10.)

Approximately 16 piloted flights were performed for each of the six combinations of off-nominal initial conditions considered in this study. Since the touchdown conditions attained when β was specified were essentially the same as those attained by using the alternate technique, the results obtained by these techniques are combined in figure 11. The pilots generally touched down with velocity components of less than 10 ft/sec (3.048 m/sec) and within about 6000 ft (1828.8 m) of the nominal landing site. The characteristic velocity required to fly the off-nominal trajectories was generally within about 10 percent of that required for a perfectly flown nominal trajectory.

The arithmetic mean and the standard deviation from the mean for touchdown conditions for both the nominal and off-nominal trajectories are presented in the following table:

Parameter	Arithmetic mean	Standard deviation
$\dot{R}$	3.49 ft/sec (1.064 m/sec)	3.76 ft/sec (1.146 m/sec)
$R\dot{\theta}$	0.84 ft/sec (0.256 m/sec)	2.77 ft/sec (0.844 m/sec)
$\dot{z}$	0.52 ft/sec (0.158 m/sec)	5.10 ft/sec (1.554 m/sec)
ΔR_a	-274 ft (-83.515 m)	3676 ft (1120.445 m)
Δz	377 ft (114.910 m)	1456 ft (443.789 m)
ΔV	6260 ft/sec (1908.048 m/sec)	210 ft/sec (64.008 m/sec)

It should be noted that the arithmetic mean pertains to the actual value of the parameter rather than to the difference between the actual and nominal values of the parameter.

CONCLUDING REMARKS

A fixed-base-simulator study has been conducted of the ability of pilots to make soft lunar landings by using a simplified guidance technique to deorbit from a synchronous transfer orbit and place the landing vehicle in a position from which a nearly vertical descent to touchdown in a specified area can be accomplished. The study included all six rigid-body degrees of freedom of the vehicle. No automatic damping or control was assumed for the vehicle. The pilot was to maintain a constant angle between the vehicle thrust vector and the line of sight to the orbiting command module until he attained nearly zero velocity at an altitude of approximately 5000 ft (1524 m). The pilot then performed a vertical descent to the lunar surface.

The results of the investigation showed that the pilot's use of the simplified guidance technique for both nominal and off-nominal trajectories resulted in placing the landing vehicle in a position from which safe landings in a specified area of the moon could consistently be made. The characteristic velocity required to perform the landings under the influence of several off-nominal initial conditions was generally within 10 percent of that required for a perfectly flown nominal trajectory.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., June 21, 1965.

APPENDIX

EQUATIONS OF MOTION

The six equations of motion governing the behavior of the vehicle are presented below. The force equations are written with respect to an inertial-axis system fixed at the center of a nonrotating spherical moon, as follows:

x-force equation,

$$\ddot{x} = \frac{F}{m}(\cos \psi \sin \theta \cos \phi + \sin \psi \sin \phi) - g_m\left(\frac{R_m}{R}\right)^2 \frac{x}{R}$$

y-force equation,

$$\ddot{y} = \frac{F}{m}(\sin \psi \sin \theta \cos \phi - \cos \psi \sin \phi) - g_m\left(\frac{R_m}{R}\right)^2 \frac{y}{R}$$

z-force equation,

$$\ddot{z} = \frac{F}{m} \cos \theta \cos \phi - g_m\left(\frac{R_m}{R}\right)^2 \frac{z}{R}$$

The moment equations are written with respect to the vehicle body axes, as follows:

X-moment equation,

$$\dot{p} = \frac{1}{I_X}\left[M_X - qr\left(I_Z - I_Y\right)\right]$$

Y-moment equation,

$$\dot{q} = \frac{1}{I_Y}\left[M_Y - pr\left(I_X - I_Z\right)\right]$$

Z-moment equation,

$$\dot{r} = \frac{1}{I_Z}\left[M_Z - pq\left(I_Y - I_X\right)\right]$$

In addition, several auxiliary equations were employed:

$$\dot{\psi} = \frac{r \cos \phi + q \sin \phi}{\cos \theta}$$

$$\dot{\theta} = q \cos \phi - r \sin \phi$$

APPENDIX

$$\dot{\phi} = p + \dot{\psi} \sin \theta$$

$$M_X = \frac{\partial M_X}{\partial \delta_X} \delta_X$$

$$M_Y = \frac{\partial M_Y}{\partial \delta_Y} \delta_Y$$

$$M_Z = \frac{\partial M_Z}{\partial \delta_Z} \delta_Z$$

$$m = m_o + \int_0^t \dot{m}\, dt$$

$$\dot{m} = \frac{F}{g_e I_{sp}}$$

$$F = - \frac{\partial F}{\partial \delta_F} \delta_F$$

$$R = \sqrt{x^2 + y^2 + z^2}$$

$$\dot{R} = \frac{x\dot{x} + y\dot{y} + z\dot{z}}{R}$$

$R\dot{\bar{\theta}} = \dot{x} \sin \bar{\theta} + \dot{y} \cos \bar{\theta}$ where:

$$\bar{\theta} = \tan^{-1}\left(\frac{-y}{x}\right)$$

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3. Mechtly, E. A.: The International System of Units - Physical Constants and Conversion Factors. NASA SP-7012, 1964.

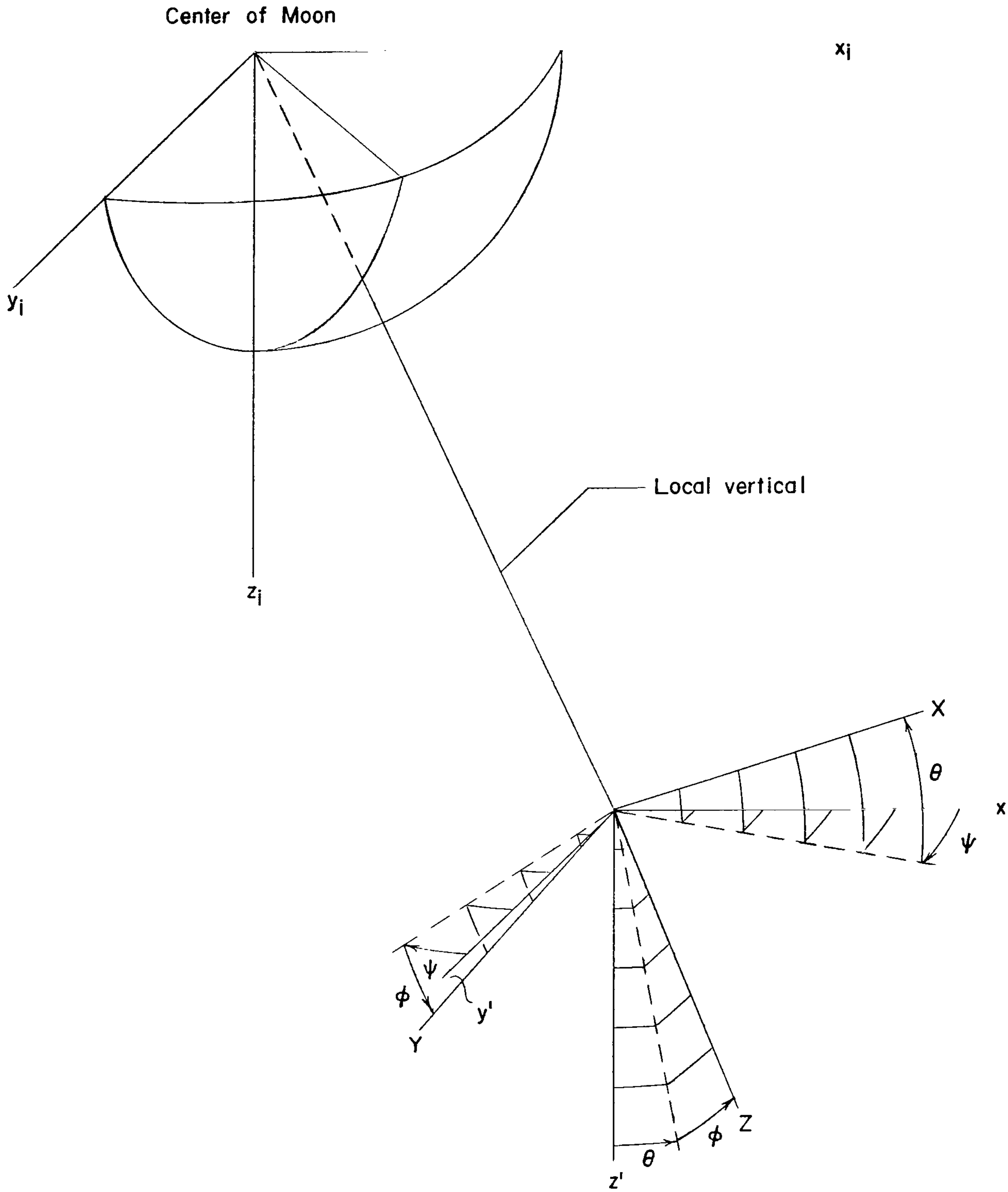


Figure 1.- Inertial-axis and vehicle-axis systems.

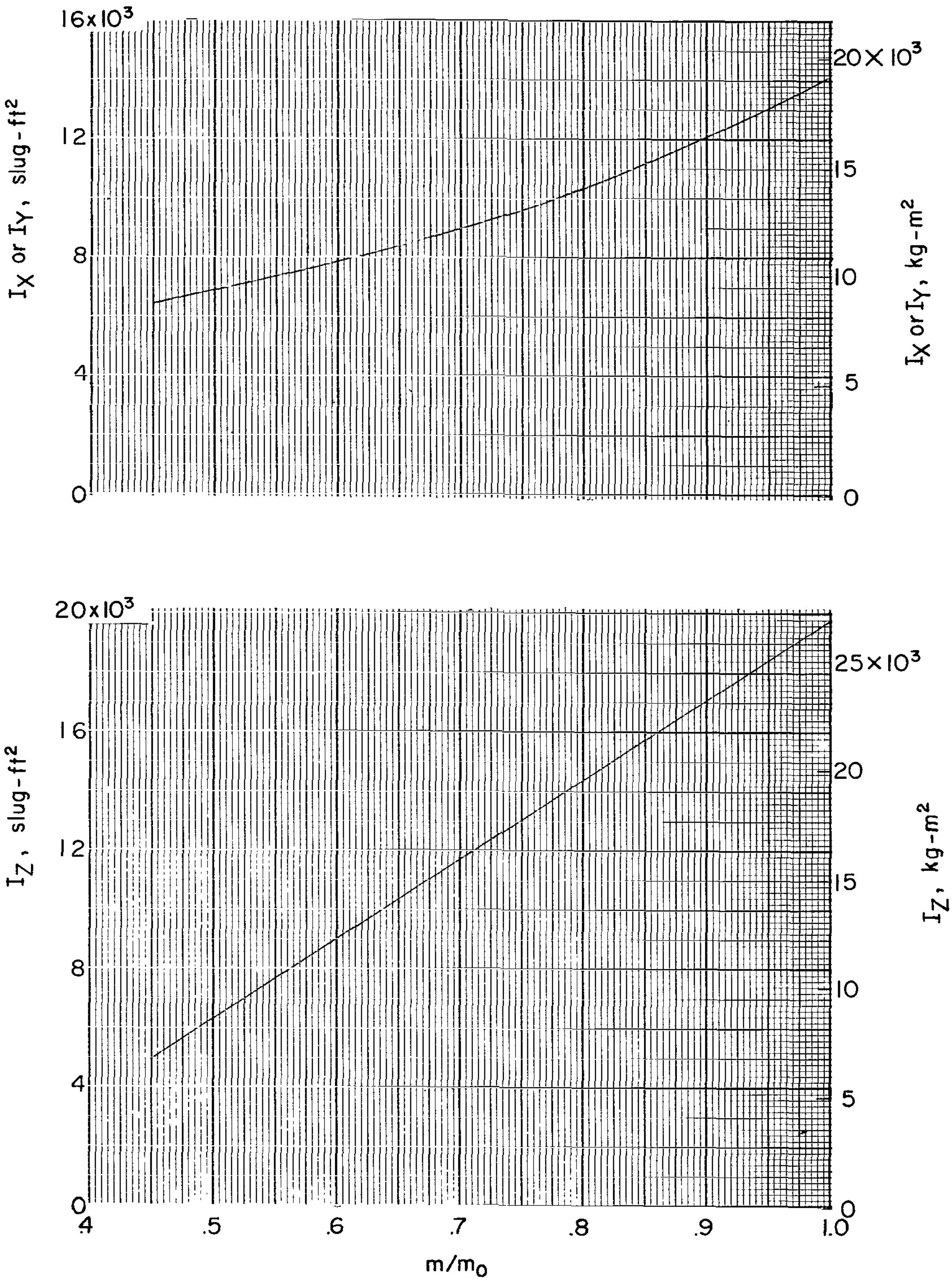


Figure 2.- Vehicle characteristics.

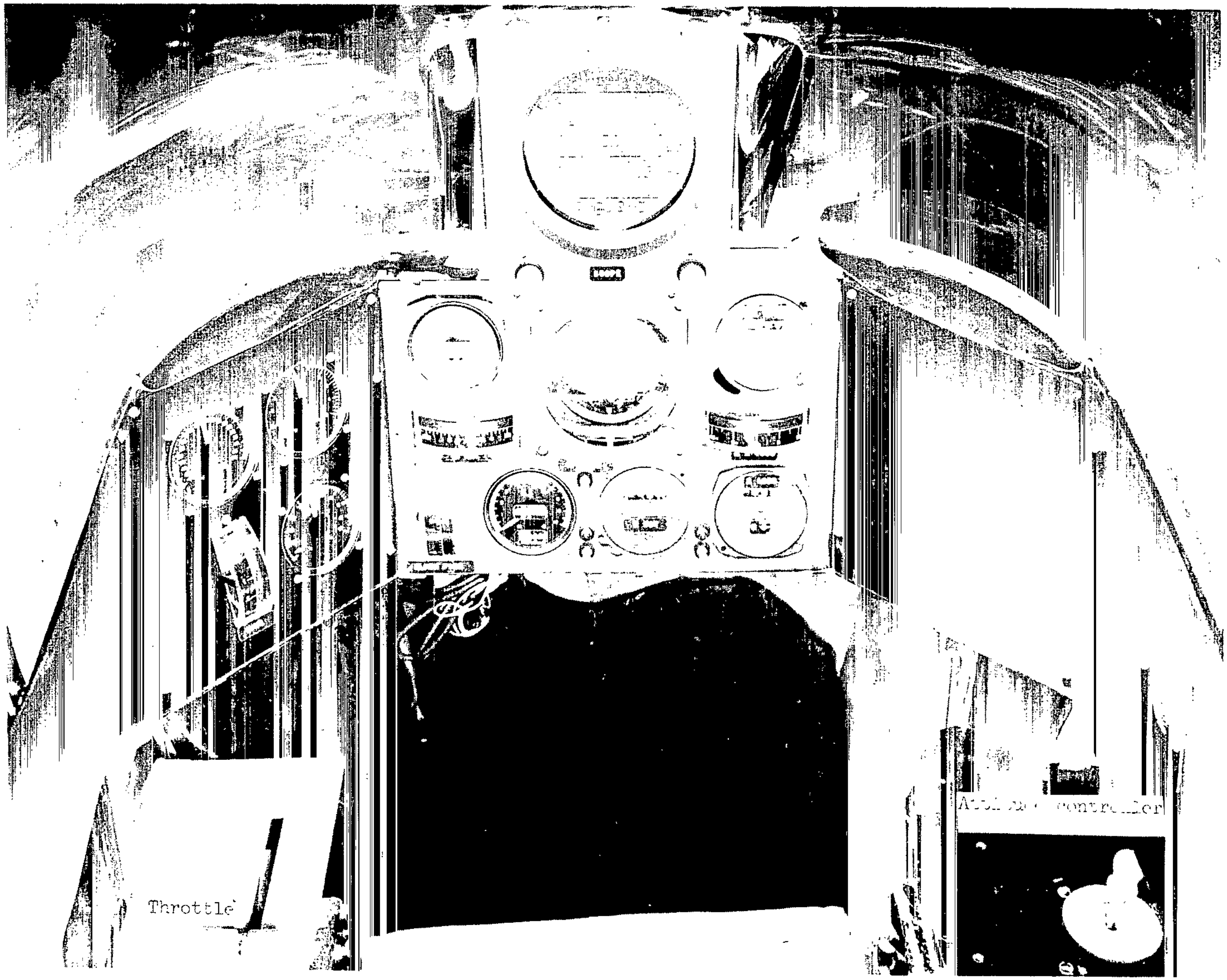


Figure 3.- General layout of cockpit.

L-64-5387.1

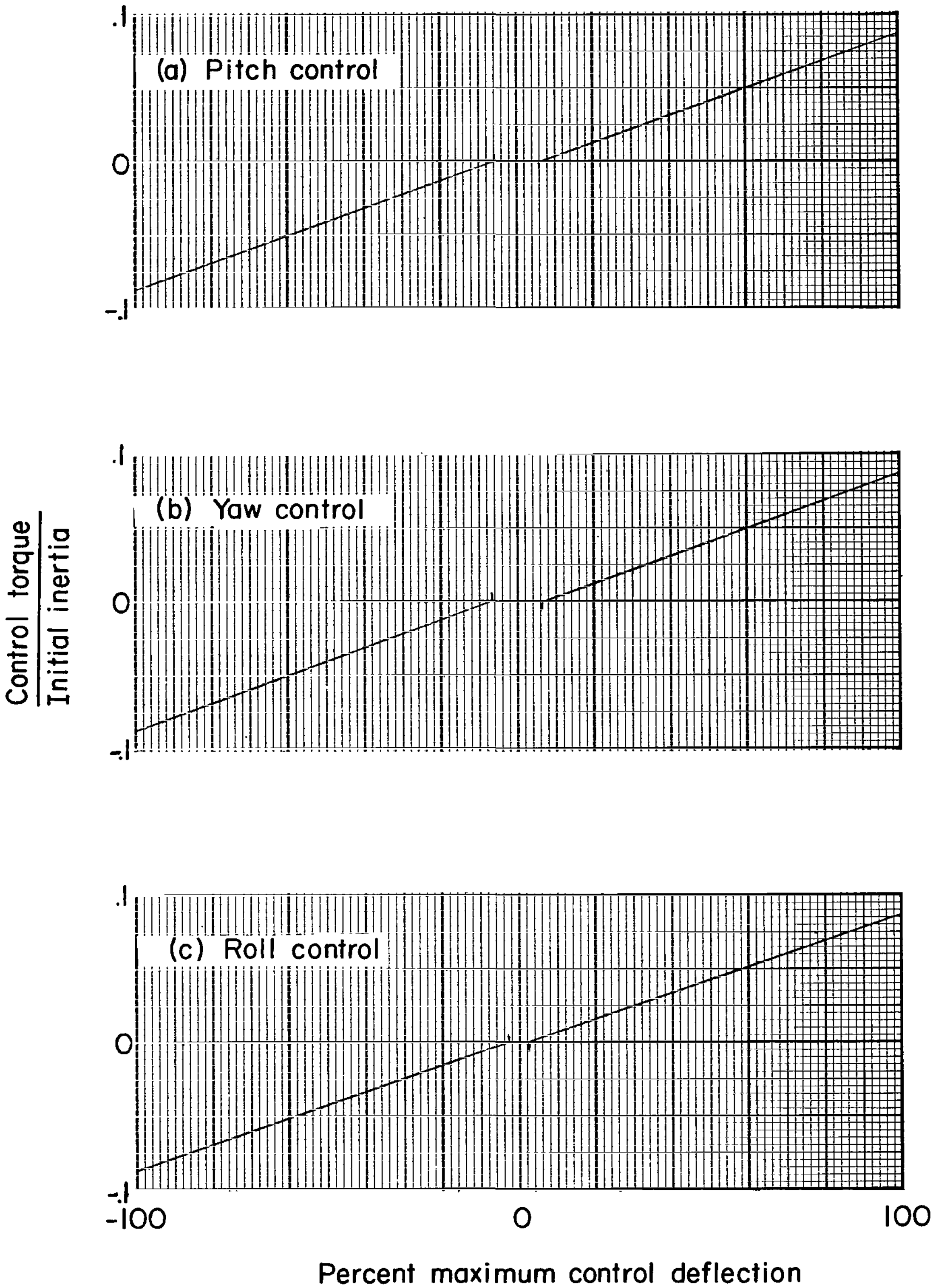


Figure 4.- Initial variation of control torque with control deflection.

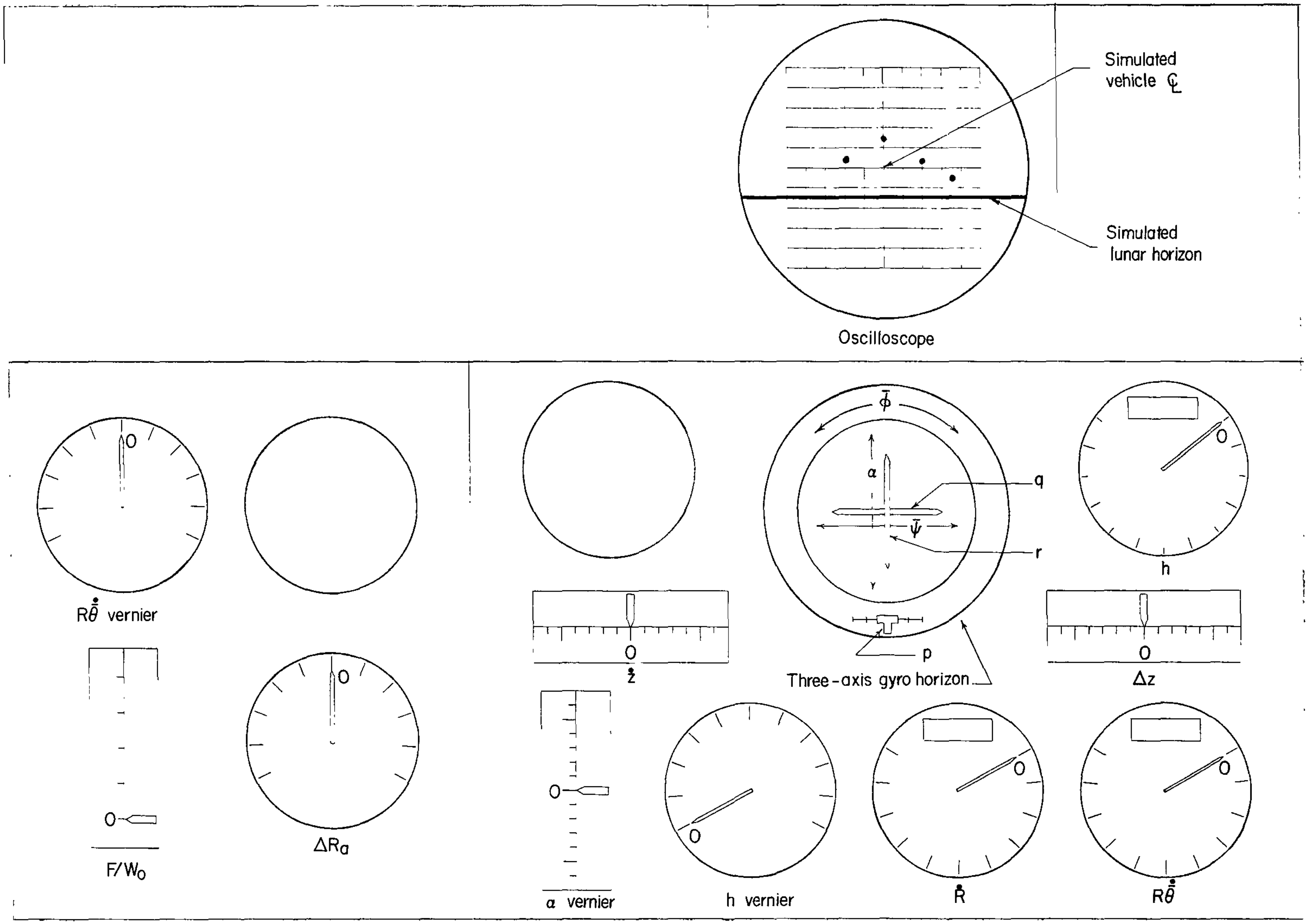


Figure 5.- Instrument panel.

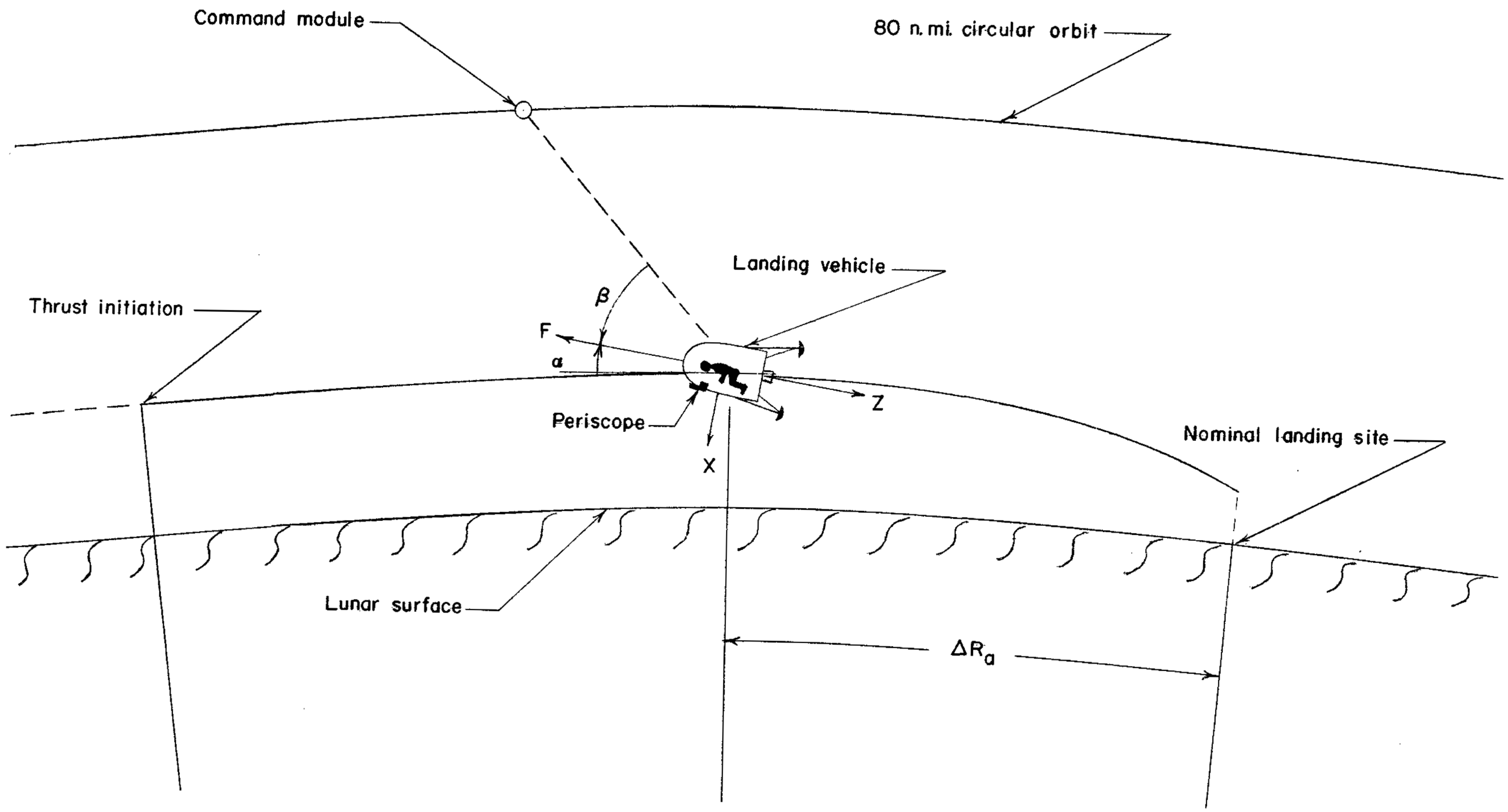
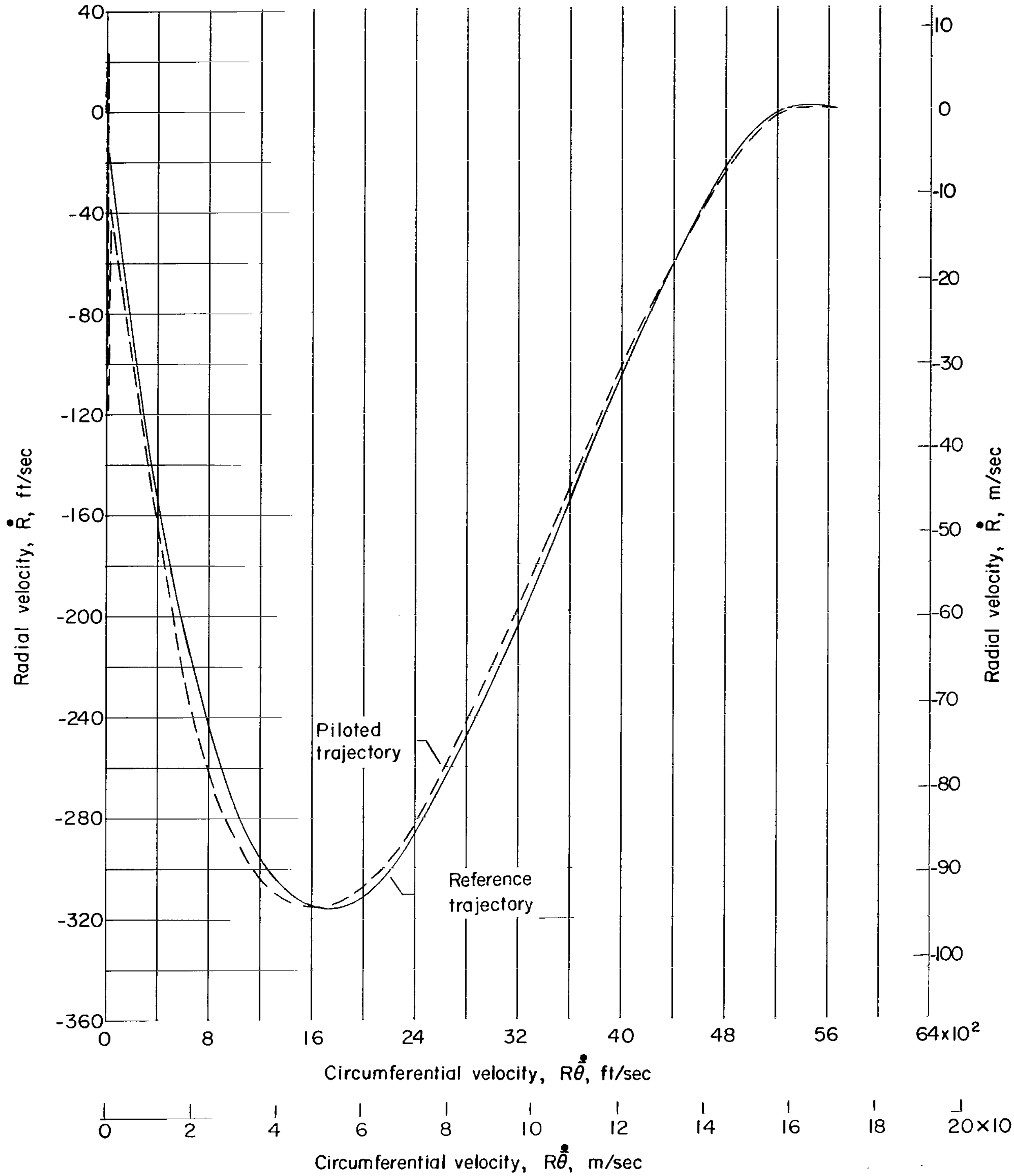
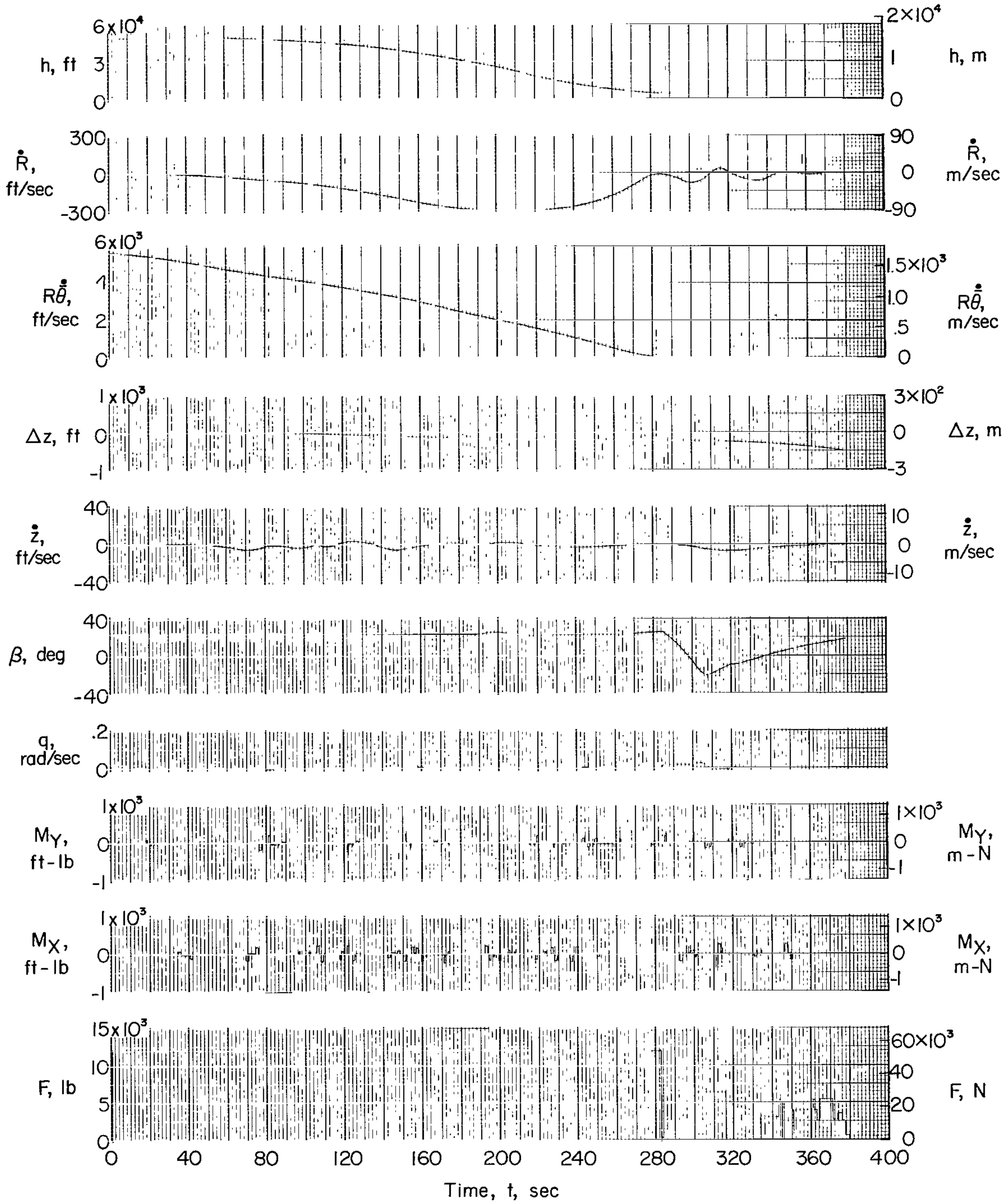


Figure 6.- Lunar landing trajectory.



(a) Variation of radial velocity with circumferential velocity.

Figure 7.- Typical flight of nominal trajectory. ΔV = 6237 ft/sec (1901.0 m/sec).



(b) Time history.

Figure 7.- Concluded.

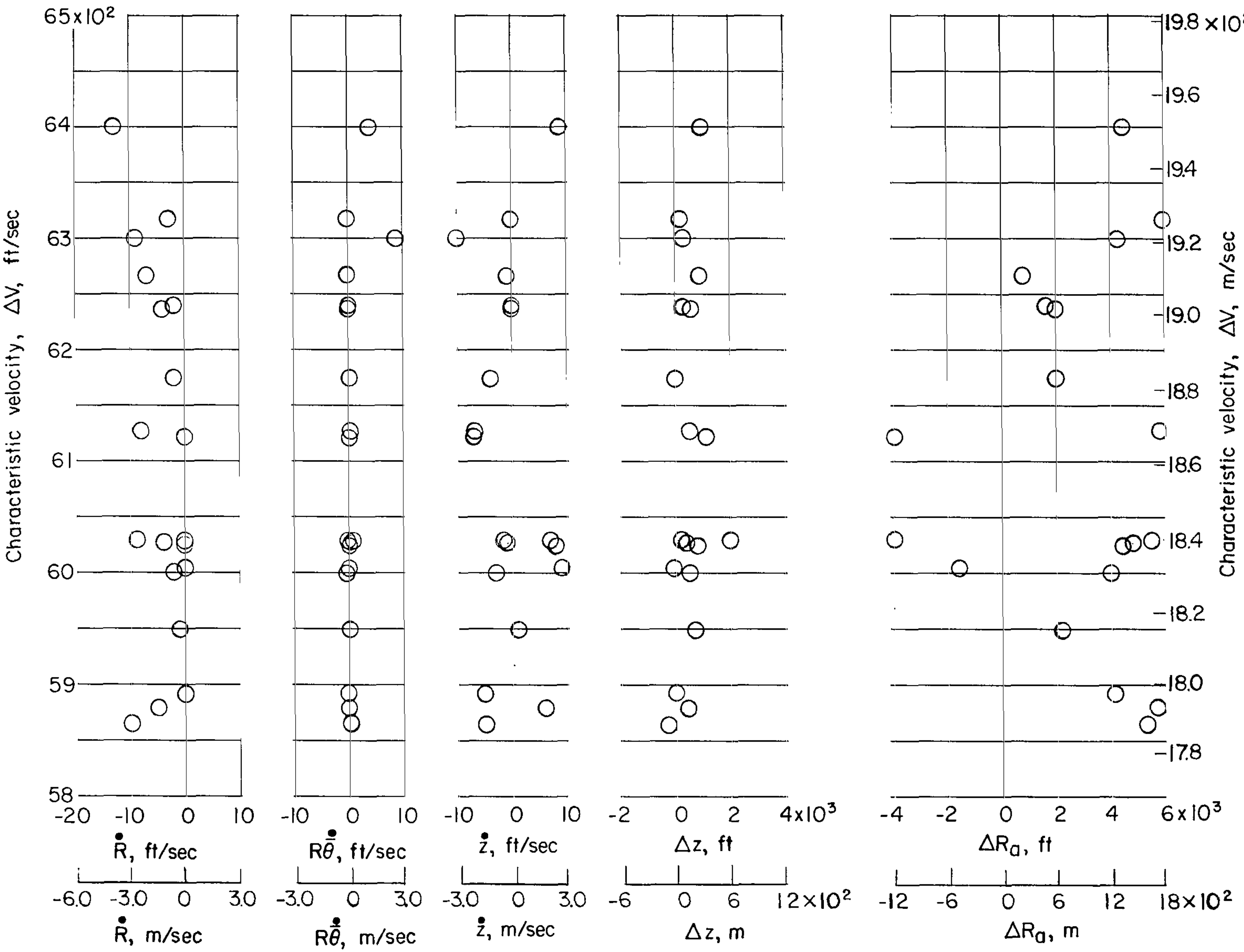
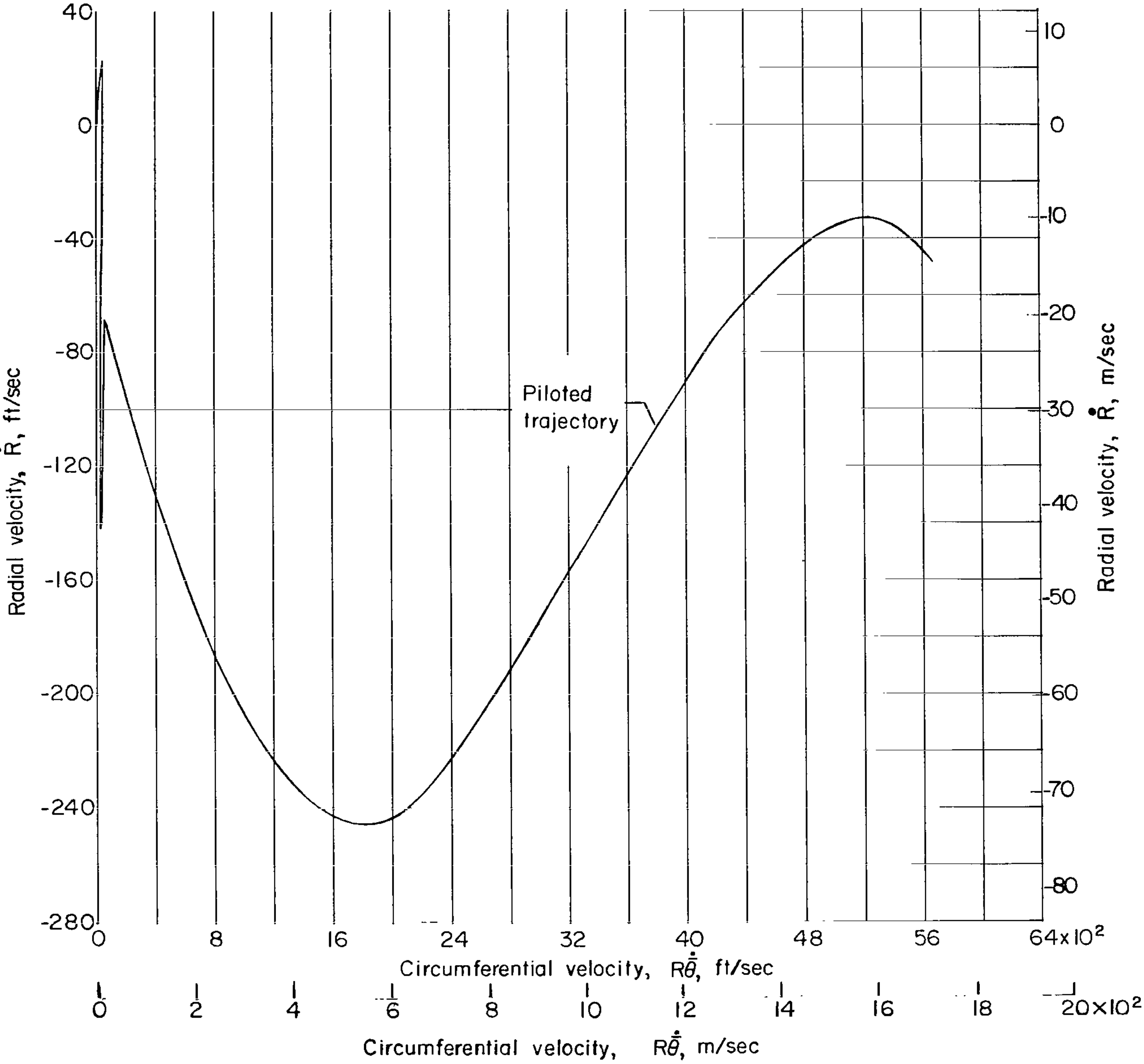
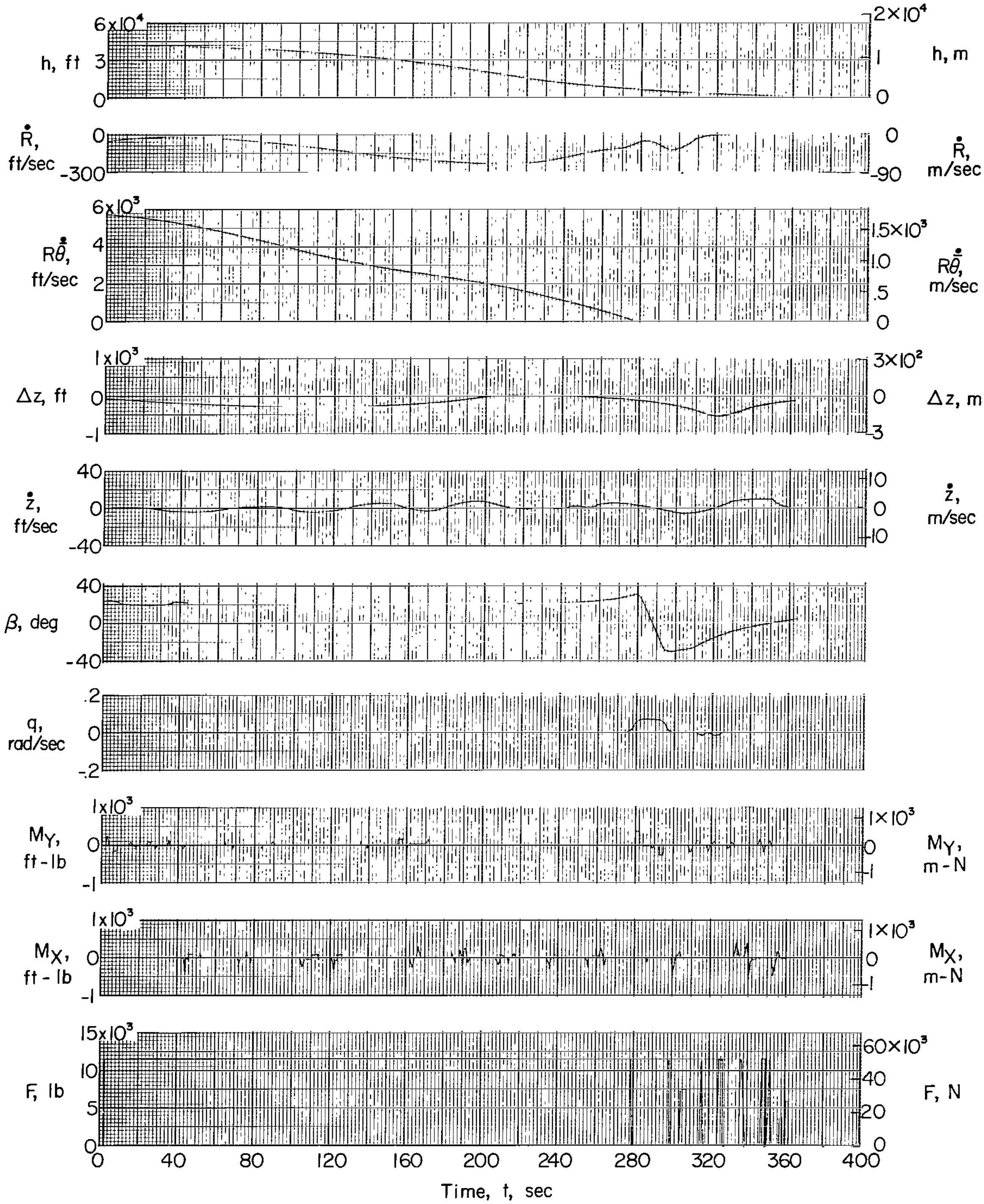


Figure 8.- Touchdown conditions of nominal flights.



(a) Variation of radial velocity with circumferential velocity.

Figure 9.- Typical flight of off-nominal trajectory for which computed β technique was used. $\dot{R}_0$ = -50 ft/sec (-15.2 m/sec); $(R\dot{\theta})_0$ = 5623 ft/sec (1713.9 m/sec); h_0 = 40 000 ft (12 192 m); ΔV = 6187 ft/sec (1885.8 m/sec).



(b) Time history.

Figure 9.- Concluded.

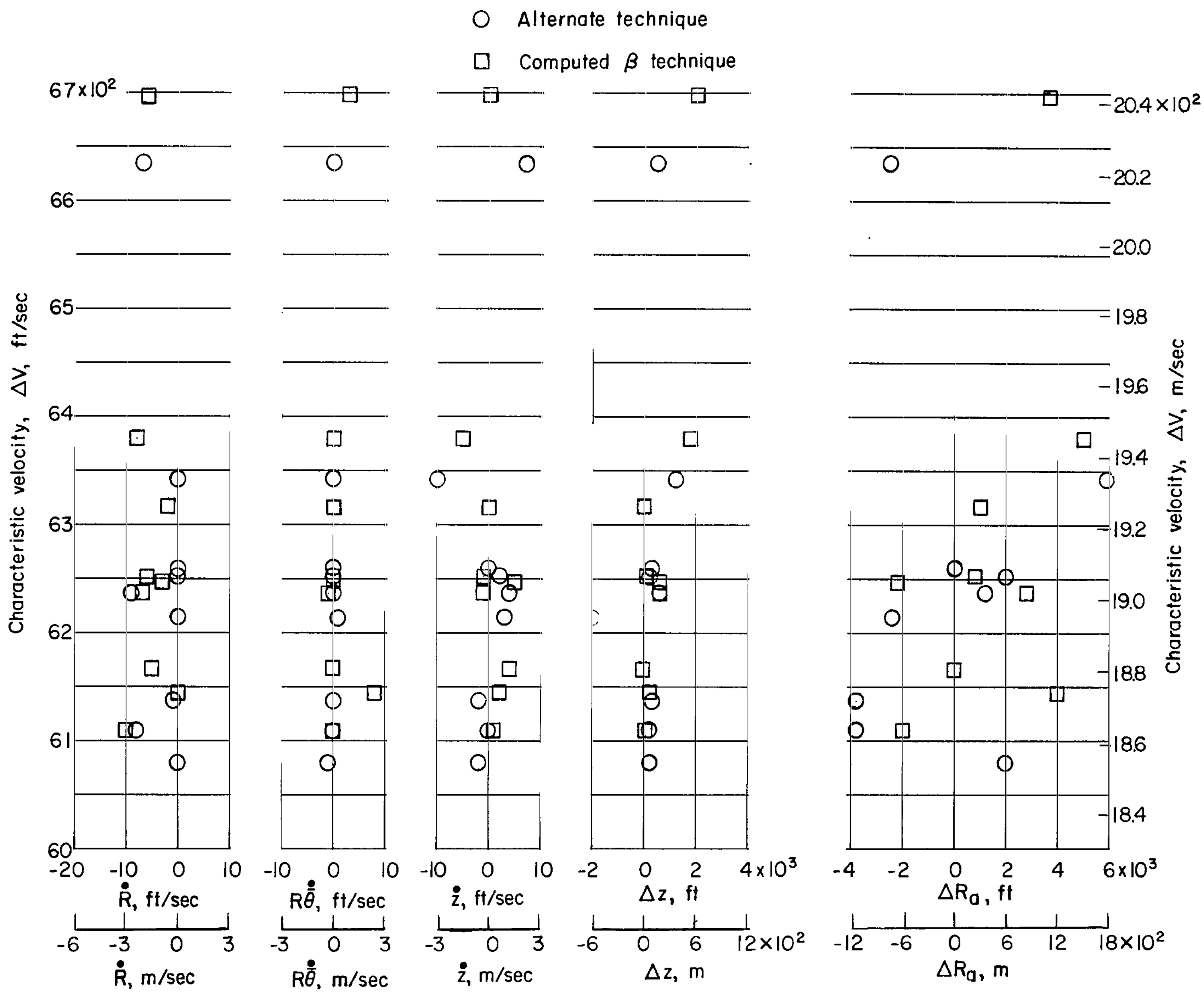


Figure 10.- Comparison of touchdown conditions attained by using the two guidance techniques in flying off-nominal trajectory. $\dot{R}_0$ = -50 ft/sec (-15.2 m/sec); $(R\dot{\theta})_0$ = 5673 ft/sec (1729.1 m/sec); h_0 = 50 000 ft (15 240 m).

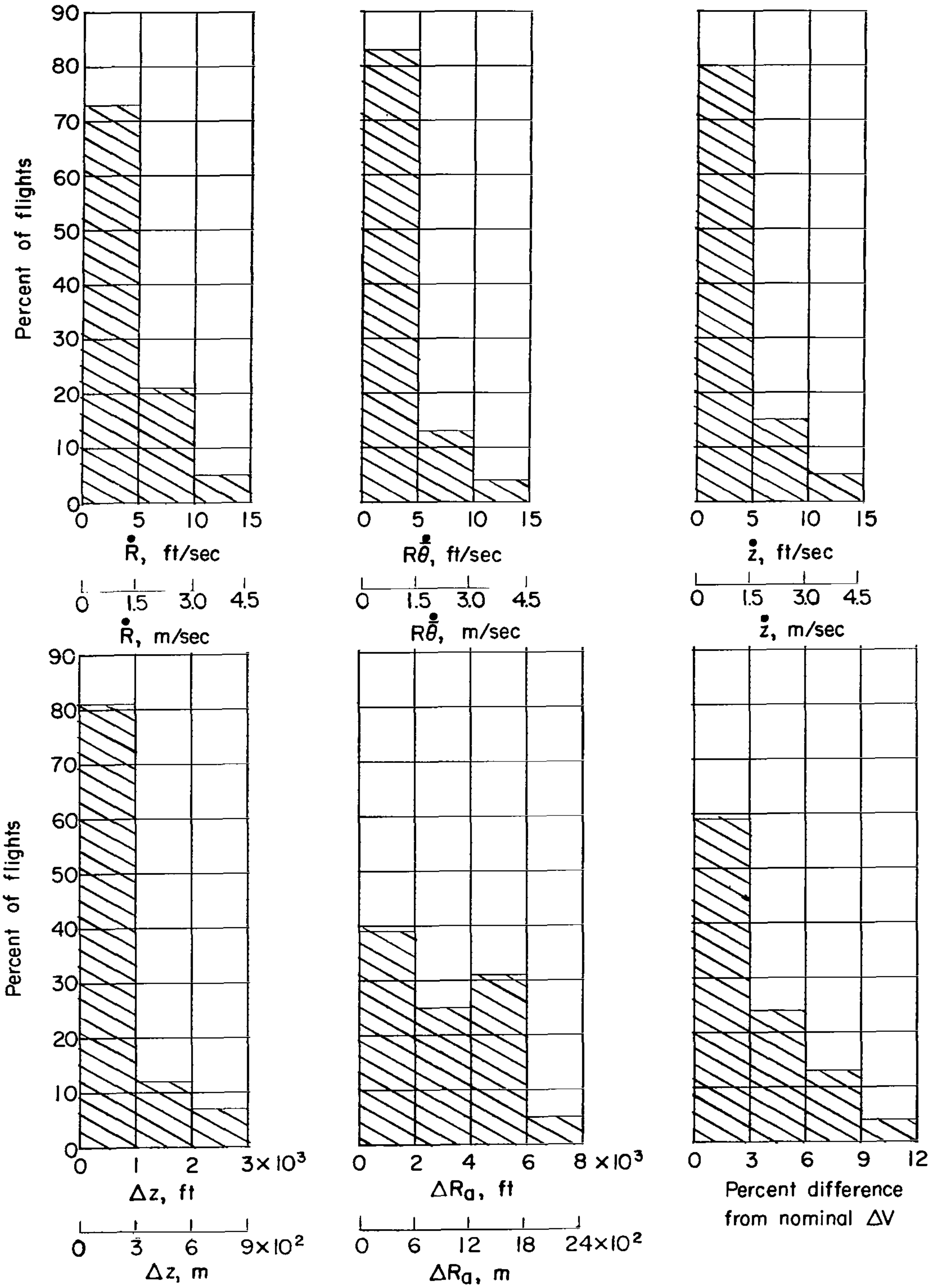


Figure 11.- Summary of touchdown conditions of off-nominal trajectories.

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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